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RESEARCH MEMORANDUM

ANALYSIS OF MEASURED PRESSURES ON AIRFOILS AT MACH NUMBERS NEAR 1

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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ANALYSIS OF MEASURED PRESSURES ON AIRFOILS AT MACH NUMBERS NEAR 1

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SUMMARY

Measured pressures over airfoils at Mach numbers near 1, with subsonic velocities at the nose and supersonic velocities throughout the rear portion, are analyzed by comparison with calculations for simpler cases. The flow directly behind the sonic point on an airfoil is of a complicated nature and is not subject to the abrupt discontinuity inherent to the Prandtl-Meyer flow at a Mach number of 1. Neither the Prandtl-Meyer flow, a purely supersonic theorem, nor the flow determined by a linear-velocity extrapolation of the pressure distribution in the subsonic region is valid throughout the region of supersonic flow over the airfoils. The Prandtl-Meyer flow, however, is shown to be an excellent approximation of the measured flow over an airfoil if applied at sufficiently high Mach numbers starting at a point at, or rearward of, the airfoil maximum thickness location. The linear-velocity extrapolation of the subsonic pressure distribution holds for only a short distance rearward of the sonic-velocity location. The computed boundary-layer thickness was found to change the turning angle of the flow very little if separation of flow from the airfoil is not considered.

INTRODUCTION

With present available theories the flow over aerodynamic surfaces may be easily computed provided that the flow is entirely subsonic or entirely supersonic. When subsonic and supersonic flows occur simultaneously over various parts of the same surface, however, calculation of the flow becomes extremely difficult. In the specific case of two-dimensional flow with a stream Mach number of 1, a solution has been obtained for an airfoil shape designed expressly for ease in computing the flow (reference 1). However, the work of reference 1 has not been extended to compute the flow for a more general airfoil.

It is thus necessary, even in the case of a stream Mach number of 1, to rely on experimental information obtained at transonic Mach numbers to analyze problems in the region of mixed flow.

During an analysis of pressure-distribution data obtained in the Langley annular transonic tunnel for an NACA 66-006 airfoil at a Mach number of 1 and an angle of attack of 0° (reference 2), the experimental pressure distribution in the supersonic portion of the flow was compared with the pressure distribution obtained from the Prandtl-Meyer pressure-turning-angle relationship (described in reference 3). The Prandtl-Meyer flow was computed rearward from the measured sonic-velocity location on the airfoil surface. It is shown in reference 2 that although the approximation indicated higher local supersonic velocities than were experimentally realized, the experimental and computed points of maximum velocity as well as the general shapes of the supersonic portion of the pressure-distribution curves were in good agreement.

The data of reference 2 are for only one airfoil at one Mach number and one angle of attack. Recently, however, additional data have been obtained for another airfoil section through a range of Mach numbers near 1 at several angles of attack. The additional data were obtained for blade sections of a rotating full-scale propeller in the Langley 16-foot high-speed tunnel (reference 4). If comparison with the Prandtl-Meyer flow is made in the same manner as in reference 2 for a blade section operating at a Mach number near 1, the same conclusions are reached as in reference 2; that is, the approximation predicts the shape of the supersonic portion of the pressure-distribution curve, but indicates higher local supersonic velocities than were measured.

As it appeared that the Prandtl-Meyer flow could be used to advantage in analyzing data obtained in the mixed-flow region, it was decided to make a more detailed comparison between experimental results and theory. A study of the flow pattern obtained indicated that better agreement between experiment and theory than that presented in reference 2 might be realized by comparing the experimental result with the pressure distribution obtained by computing the Prandtl-Meyer flow from a station corresponding to a local Mach number appreciably higher than 1 and by using a polynomial extrapolation in the proximity of the sonic line.

In the present paper, the measurements upon which references 2 and 4 are based are utilized to make further comparisons with theory. It is not the purpose of this paper to present an empirical method from which pressure distributions may be estimated, but to compare the pressures measured in the supersonic region of the transonic flow over several airfoils with pressures computed for simpler instances. In addition, the effect of the airfoil boundary layer on the turning angle of the flow is studied.

DISCUSSION

Theoretical basis of approximations.– The purely supersonic flow past an airfoil may be computed as a simple Prandtl–Meyer flow (reference 3). The flow is assumed parallel to the airfoil surface and is represented by a series of Mach waves originating at the surface. Only one family of the two possible families of characteristics need be considered. With transonic flow, however, the supersonic flow directly behind the sonic line may not be truly represented by the one–family characteristic system of the Prandtl–Meyer flow. Expansion waves originating at the airfoil surface reach the sonic–velocity line and cause compression waves which return to the airfoil surface. Thus the general double–wave structure or characteristic system is the true representation in this region (references 1 and 5).

At the sonic velocity, the Prandtl–Meyer flow has an infinite rate of change in velocity (or pressure) with turning angle. The theory of references 1 and 5, however, indicates that the flow past the sonic–velocity location on an airfoil of analytic contour is not subject to the abrupt discontinuity inherent to the Prandtl–Meyer flow. The Prandtl–Meyer flow, neglecting the incoming compressions, indicates the lower limit of the pressures over an airfoil surface in transonic flow rather than the actual pressure distribution.

At stream Mach numbers near 1, the sonic–velocity line after starting as usual at an oblique angle turns more and more perpendicular to the streamlines. The flow field, therefore, becomes more parallel to the surface and thus better satisfies one of the requirements of the Prandtl–Meyer flow. The effect is a better approximation of the pressure distribution if the Prandtl–Meyer flow is computed from a point farther downstream than the sonic–velocity location.

According to the indications of references 1 and 5, the flow should pass smoothly through the sonic velocity and it should be possible by analytic continuation of the pressures in the subsonic region – that is, by a purely mathematical treatment – to determine the pressures for a finite chordwise distance into the supersonic region. A simple linear extrapolation of the velocity through the sonic–velocity boundary, however, is a rather inadequate treatment, as will be shown.

<u>Comparison of Prandtl–Meyer flow with measurements</u>.– The experimental pressure distribution about an NACA 66–006 airfoil at a Mach number of 1, a Reynolds number of about 2.7×10^6, and an angle of attack of 0° (reference 2) is presented in figure 1. It is seen from the figure that the Prandtl–Meyer flow computed from the sonic–velocity location (18–percent–chord station) indicates higher local supersonic velocities over the airfoil than were experimentally obtained.

In figure 2 a similar comparison is made between experimental results and the Prandtl–Meyer flow for another airfoil section. The experimental data were obtained by measuring the pressure distribution over one blade of a two–blade propeller operated in the Langley 16–foot high–speed tunnel (reference 4). The tests were conducted with NACA 10–(3)(08)–03 design blades and the data presented were obtained at the 0.8 radius station for a section Mach number of 1.00, a Reynolds number of about 3.5×10^6, and a section angle of attack of 0.35°. The angle of attack is corrected for induced flow through the propeller. The airfoil section at this station is nearly an NACA 16–307 section. It may be seen as shown in figure 1 that the pressure distribution obtained by computing the Prandtl–Meyer flow indicates higher local supersonic velocities over the upper surface of the airfoil than those experimentally realized. Again, however, the general shapes of the experimental and theoretical curves are in good agreement.

In figure 3 the pressure distribution over the upper surface of the same section as for figure 2 is presented for a section Mach number of 1.09, a Reynolds number of about 3.8×10^6, and a section angle of attack of 3.0°. A comparison similar to figure 2 is shown and it may be noted that the difference in magnitude between the experimental velocities and those obtained by computing the Prandtl–Meyer flow is greater for an angle of attack of 3.0° than for an angle of attack of 0.35°.

As previously stated, it was believed that the Prandtl–Meyer flow would yield results closer to the measured pressures if it were computed from a point farther rearward on the airfoil than the sonic–velocity location. Accordingly, the calculations were repeated, starting with the measured pressures at the 50–percent–chord station on the airfoils and applying the Prandtl–Meyer flow both forward and rearward of this station.

The experimental and computed pressure distributions are compared for the NACA 66–006 airfoil in figure 4. The agreement of the two curves is excellent between the 35– and 60–percent–chord stations. Rearward of the 60–percent–chord station, the computed curve predicts higher velocities than those measured. Similar comparisons are made for the NACA 16-307 airfoil section in figures 5 and 6. For both cases (0.35° angle of attack and 3.0° angle of attack, respectively), the agreement between the experimental curve and the computed curve is excellent from about the 40-percent-chord station to the rear of the airfoil section.

The excellent agreement shown in figures 5 and 6 is rather surprising when it is considered that the Prandtl-Meyer flow was developed for the two-dimensional flow case and that the experimental data were obtained at Mach numbers near 1 at the 0.8 radius station of a propeller blade where the flow would not be expected to be entirely two dimensional (reference 4). For the case of the NACA 66-006 airfoil, if the Prandtl-Meyer pressure-turning-angle relationship is used in the reverse procedure to compute the airfoil profile which would theoretically produce the experimental pressure distribution obtained from the 50-percent-chord station rearward, it is found that the computed airfoil profile is thicker than the NACA 66-006 airfoil from approximately the 60-percent-chord station rearward. A rapid increase in the rate of thickening of the boundary layer at approximately the 60-percent-chord station would cause such an apparent thickening of the airfoil profile.

As a matter of interest regarding the NACA 66-006 airfoil, the Prandtl-Meyer flow was computed forward and rearward of the 65-percent-chord station to determine how far rearward it was necessary to start the application of the Prandtl-Meyer flow to obtain agreement over the rearmost portion of the airfoil. This result is presented in figure 7. It is noted that the agreement between the Prandtl-Meyer flow and experiment is excellent (when consideration is given to the boundary layer as discussed later) from the 65-percent-chord station to at least the 90-percent-chord station. A series of weak shock waves probably occurs near the trailing edge of the airfoil because of the cusp, thus causing the computed curve to disagree slightly with the experimental curve from the 90-percent-chord station rearward.

In order to get an indication of the minimum free-stream Mach number at which the Prandtl-Meyer flow indicates accurately the pressure distribution from the central portion of the airfoil rearward, the pressure distribution was computed for the NACA 16-307 airfoil section for Mach numbers of 0.984, 0.955, and 0.882 by assuming agreement between experiment and theory at the 50-percent-chord station. Comparisons between experiment and the Prandtl-Meyer flow for these conditions are presented in figures 8 to 10.

It may be noted in figure 8 that when the shock wave is well toward the rear of the airfoil, good agreement may be expected between experiment and the Prandtl-Meyer flow when applied as previously stated. Figures 9 and 10 indicate that as the free-stream Mach number is reduced and the shock wave moves forward on the airfoil, the agreement between the Prandtl-Meyer flow and experiment becomes poorer. The experimental results indicate that the shock wave is at approximately the same location (slightly behind the 80-percent-chord station) for a Mach number of 0.955 at an angle of attack of 2.71° and a Mach number of 0.882 at an angle of attack of -0.54°. It is thus indicated that aside from the effect of shock-wave location (chordwise extent of supersonic region) the discrepancy between experiment and the Prandtl-Meyer flow is influenced considerably by the free-stream Mach number.

It appears possible from the foregoing curves that near a stream Mach number of 1 the Prandtl-Meyer flow may be applied with excellent results on these airfoils from the central part of the airfoil rearward. As the Mach number is reduced from 1, however, the accuracy of the Prandtl-Meyer flow is reduced. Also, if the boundary layer thickens appreciably or separates, the airfoil surface may no longer be considered the boundary of supersonic flow. It is evident that the flow phenomenon directly aft of the sonic-velocity location is of a complicated nature and cannot be accurately calculated with the simple Prandtl-Meyer flow.

Comparison of linear-velocity extrapolation with measurements.- It would have been interesting, with sufficiently complete and accurate data in the subsonic region of the flow about an airfoil at Mach numbers near 1, to investigate the extent to which an analytic extrapolation of the velocity into the local supersonic region is reasonably valid. Unfortunately, as can be seen in the figures, the number of data points in the subsonic region was generally quite inadequate for such a study. With the given data, little more than linear-velocity extrapolations of the velocity through the sonic point (similar to the procedure of reference 6) are justified. Such extrapolations are shown in figures 11 to 13. It may be noted that the extrapolated curves and the experimental curves are in agreement for some finite chordwise distance rear of the sonic-velocity location, thus indicating that the flow through the sonic velocity is continuous. The region of agreement, however, is so small that a linear-velocity extrapolation is of little value in predicting the local supersonic flow over the surfaces considered.

Effect of boundary layer upon airfoil pressures.- The presence of a boundary layer on an airfoil effectively increases the thickness of the airfoil profile and thus affects the airfoil pressures. Consequently, it is important to evaluate the magnitude of this effective thickening. Calculations of the boundary-layer displacement thickness were therefore made for the NACA 66-006 and NACA 16-307 airfoils to obtain the boundaries of air flow from which pressures were calculated with the Prandtl-Meyer flow.

The boundary-layer-thickness calculations were based upon the measured airfoil pressures and were made with the compressible-flow equations presented in reference 7. As the location of transition from a laminar to a turbulent boundary layer was not known, assumptions were made regarding this point. For the NACA 16-307 airfoil, transition was assumed to occur at the 50-percent-chord station. For the NACA 66-006 airfoil, the transition was assumed at the 60-percent-chord station in order to determine whether or not transition was the cause of the difference, shown in figure 4 to start at approximately this station, between experiment and the Prandtl-Meyer flow.

For the estimation of the displacement thicknesses of the laminar and turbulent boundary layers, the theoretical curves of reference 7 were used. These curves gave values for the ratio of displacement thickness to momentum thickness of about 3.6 for the laminar layer and about 1.9 for the turbulent layer (for which values of H_1, in the notation of reference 8, were taken as 2.59 and 1.2, respectively).

Shown in figure 14 are the computed growths of the boundary-layer momentum and displacement thicknesses along the surface of the NACA 66-006 airfoil with transition assumed at the 60-percent-chord station. As the displacement thickness was determined by computing the momentum thickness and applying the ratio of displacement thickness to momentum thickness, a discontinuity occurs in the displacement-thickness curve at the point of transition. This discontinuity occurs because the ratio of displacement thickness to momentum thickness is assumed to change from 3.68 to 1.85 as the boundary layer changes from laminar to turbulent.

The effect of the computed boundary-layer thickness upon the airfoil pressures is shown in figures 15 and 16. It is indicated that the computed boundary-layer thickness has a small effect upon the airfoil pressures calculated with the Prandtl-Meyer flow from the sonic points rearward. The effect is so small as to be negligible for the NACA 16-307 airfoil section and for all portions of the NACA 66-006 airfoil section except at the cusp near the trailing edge where the pressure gradient is adverse. It should be remembered that the theory of reference 7 does not consider separation or abrupt thickening of the boundary layer.

CONCLUSIONS

Comparisons of calculated pressures with measured pressures on airfoils at Mach numbers near 1 indicate the following conclusions:

1. The flow directly behind the sonic point on an airfoil is of a complicated nature and is not subject to the abrupt discontinuity inherent to the Prandtl-Meyer flow at a Mach number of 1. There is some finite distance directly behind the sonic-velocity location where the linear-velocity extrapolation of the subsonic pressures is in close agreement with the experimental pressure distribution, thus indicating that the flow through the sonic velocity is continuous.

2. Neither the Prandtl-Meyer flow, nor the flow determined by a linear-velocity extrapolation of the pressure distribution in the subsonic region is valid throughout the region of supersonic flow over the airfoils. It appears possible, however, that the Prandtl-Meyer flow may be applied at free-stream Mach numbers near or above 1 with excellent results on these airfoils from the central portion rearward.

3. As the free-stream Mach number is reduced from 1 and the shock wave at the rear of the airfoil surface moves forward, the accuracy with which the Prandtl-Meyer flow predicts the pressures from the central portion of the airfoil rearward becomes poorer. It is indicated that aside from the effect of shock-wave location (chordwise extent of supersonic region) the discrepancy between experiment and the Prandtl-Meyer flow is influenced considerably by the free-stream Mach number.

4. The computed boundary-layer thickness has a very small effect upon the airfoil pressures calculated with the Prandtl-Meyer flow if separation of flow from the airfoil is not considered.

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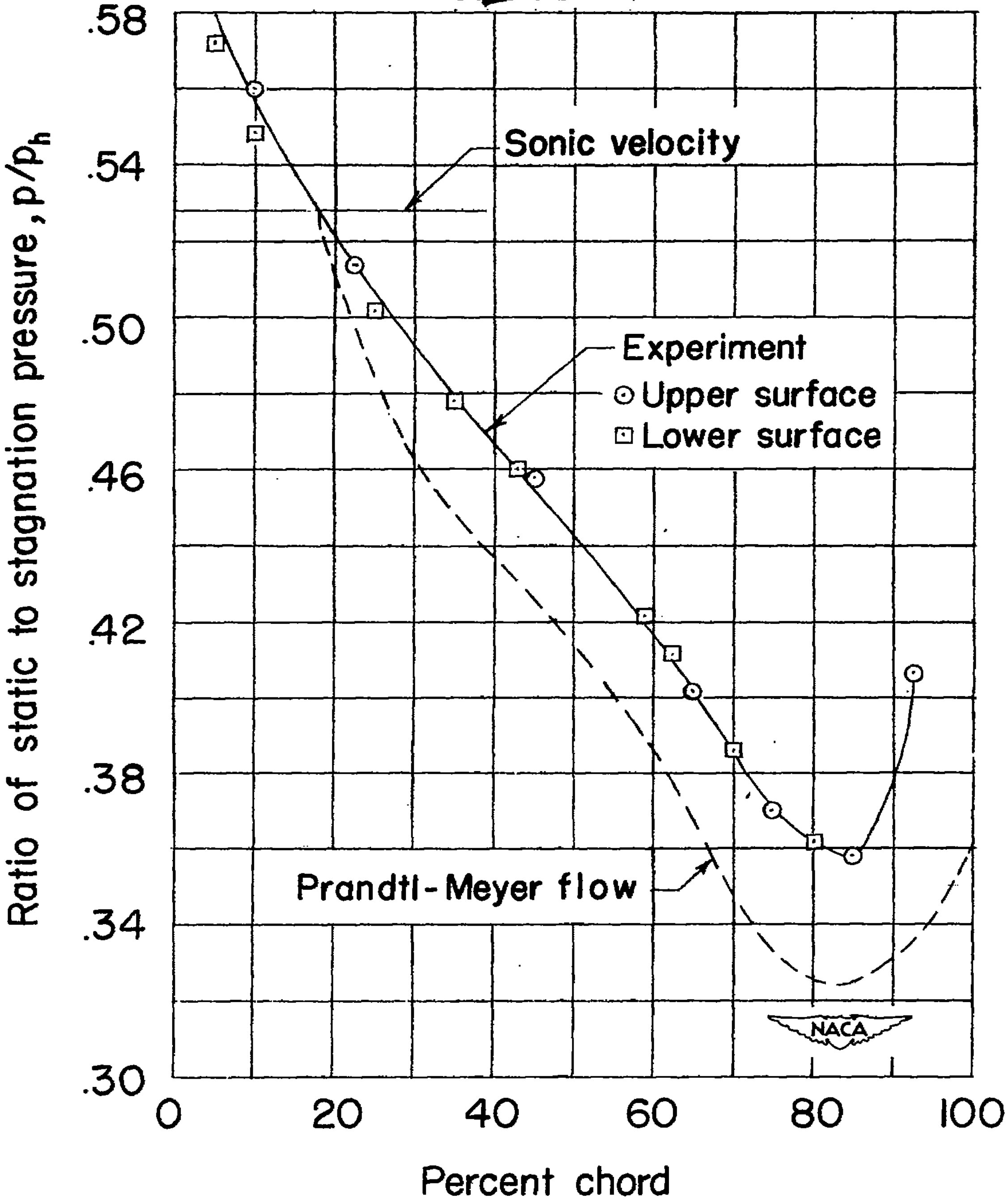

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Figure 1.- Comparison of experimental data with Prandtl-Meyer flow applied from experimentally determined sonic velocity location. NACA 66-006 airfoil; $M = 1.00$; $\alpha = 0^{\circ}$.

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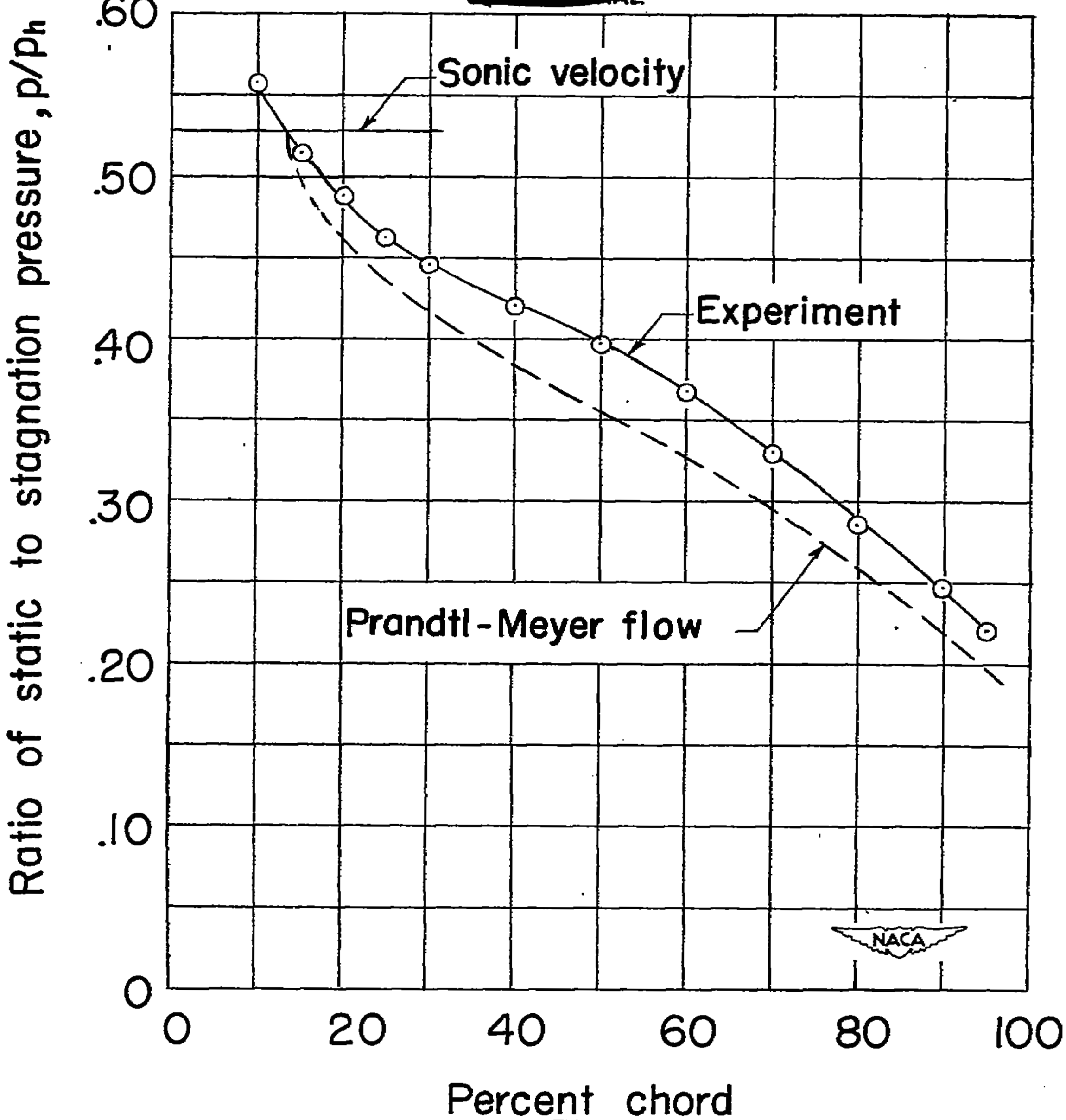

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Figure 2.- Comparison of experimental data with Prandtl-Meyer flow applied from experimentally determined sonic velocity location. NACA 16-307 airfoil; $M = 1.00$; $\alpha = 0.35^{\circ}$.

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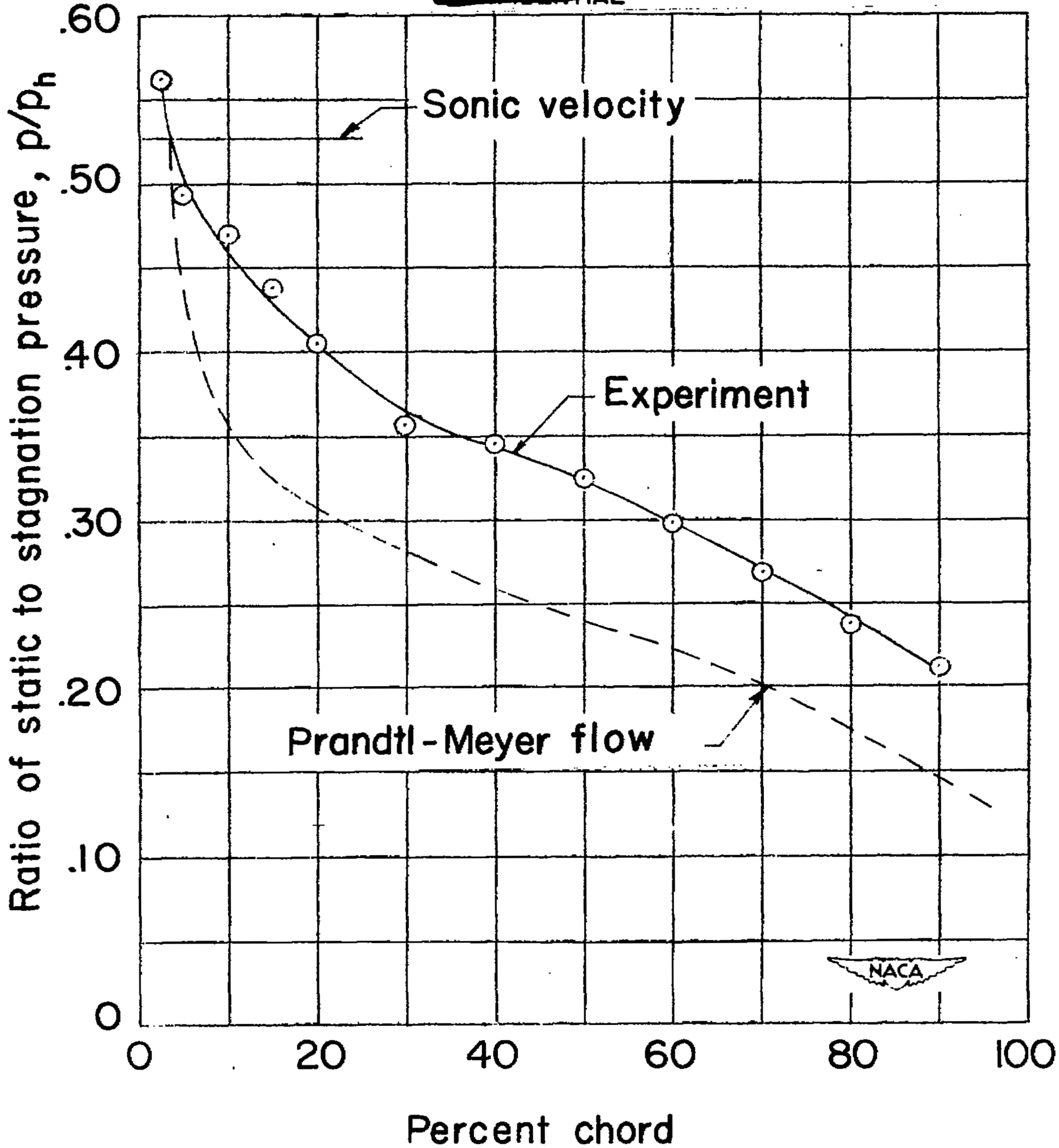

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Figure 3.- Comparison of experimental data with Prandtl-Meyer flow applied from experimentally determined sonic velocity location. NACA 16-307 airfoil; $M = 1.09$; $\alpha = 3.0^{\circ}$.

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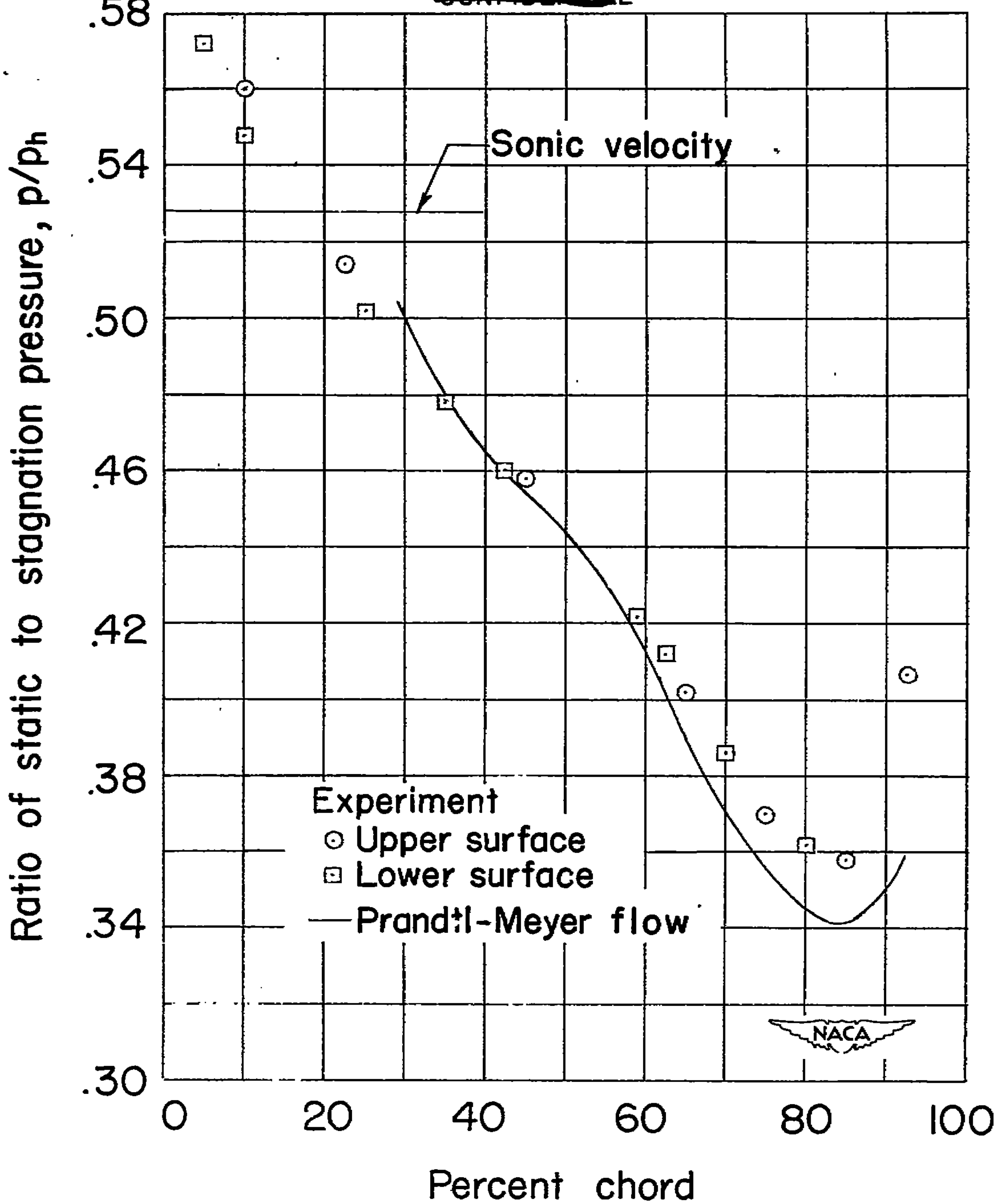

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Figure 4.- Comparison of experimental data with Prandtl-Meyer flow applied from 50-percent-chord station. NACA 66-006 airfoil; $M = 1.00$; $\alpha = 0^{\circ}$.

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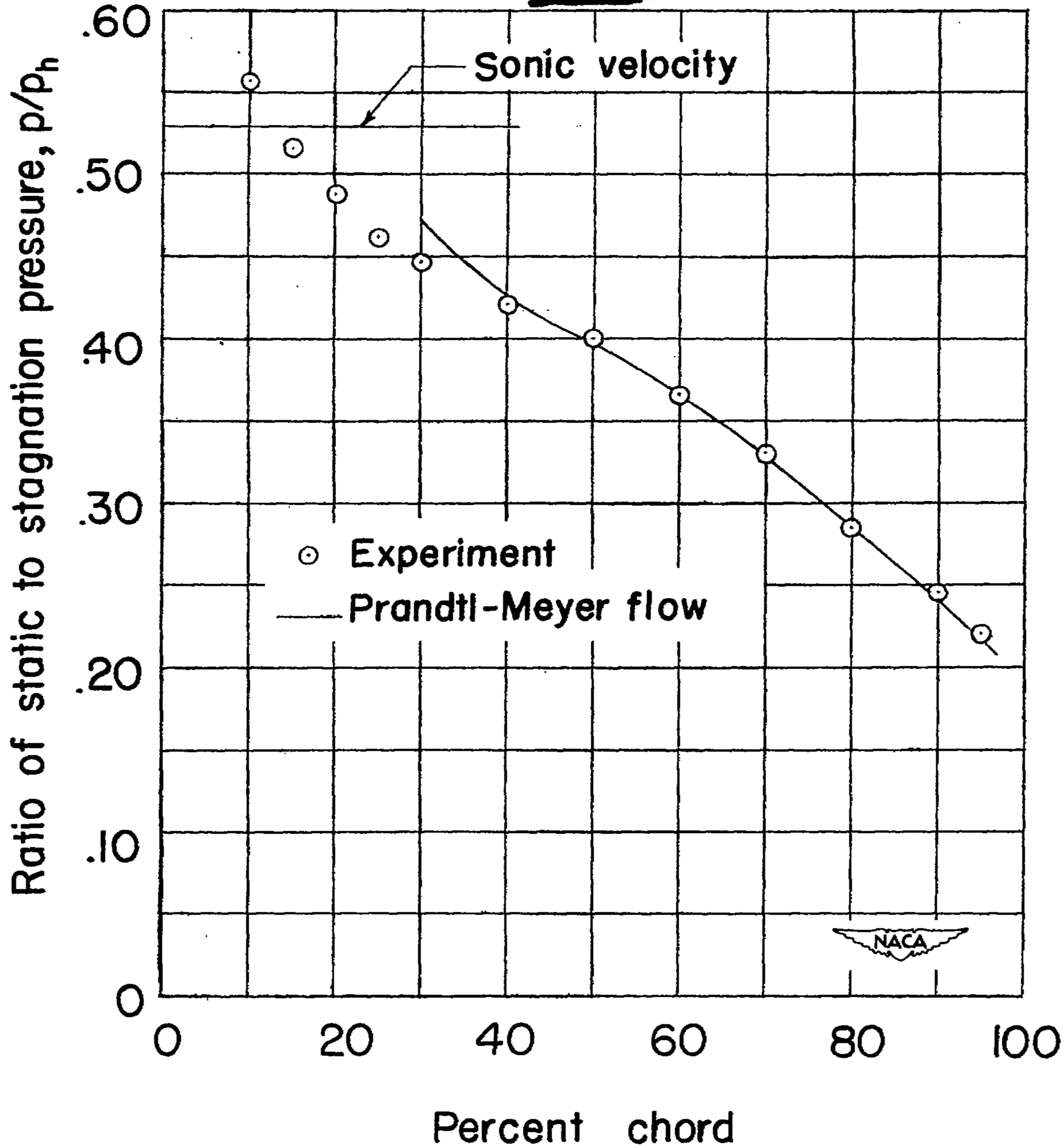

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Figure 5.- Comparison of experimental data with Prandtl-Meyer flow applied from 50-percent-chord station. NACA 16-307 airfoil; $M = 1.00$; $\alpha = 0.35^{\circ}$.

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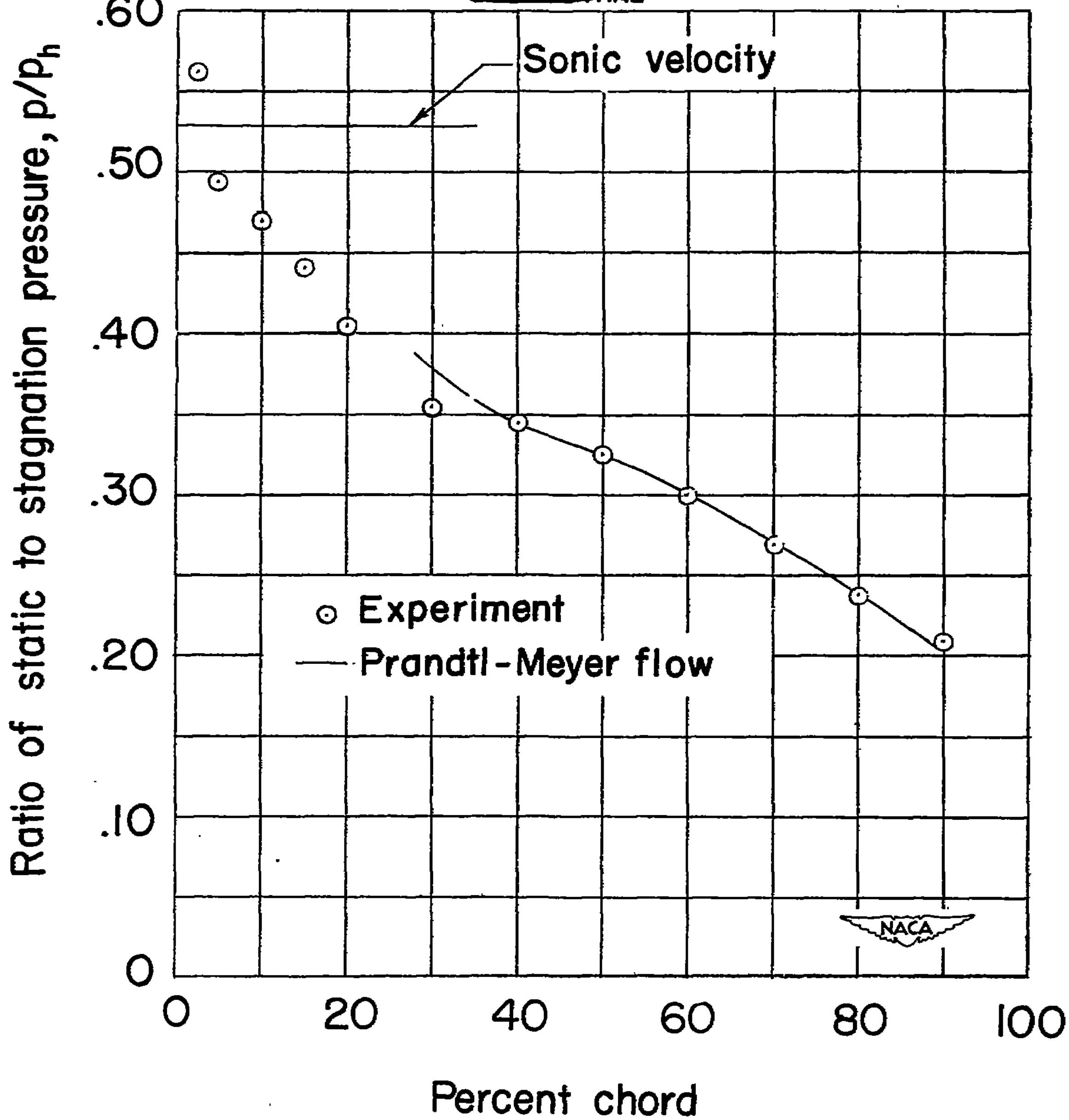

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Figure 6.- Comparison of experimental data with Prandtl-Meyer flow applied from 50-percent-chord station. NACA 16-307 airfoil; $M = 1.09$; $\alpha = 3.0^{\circ}$.

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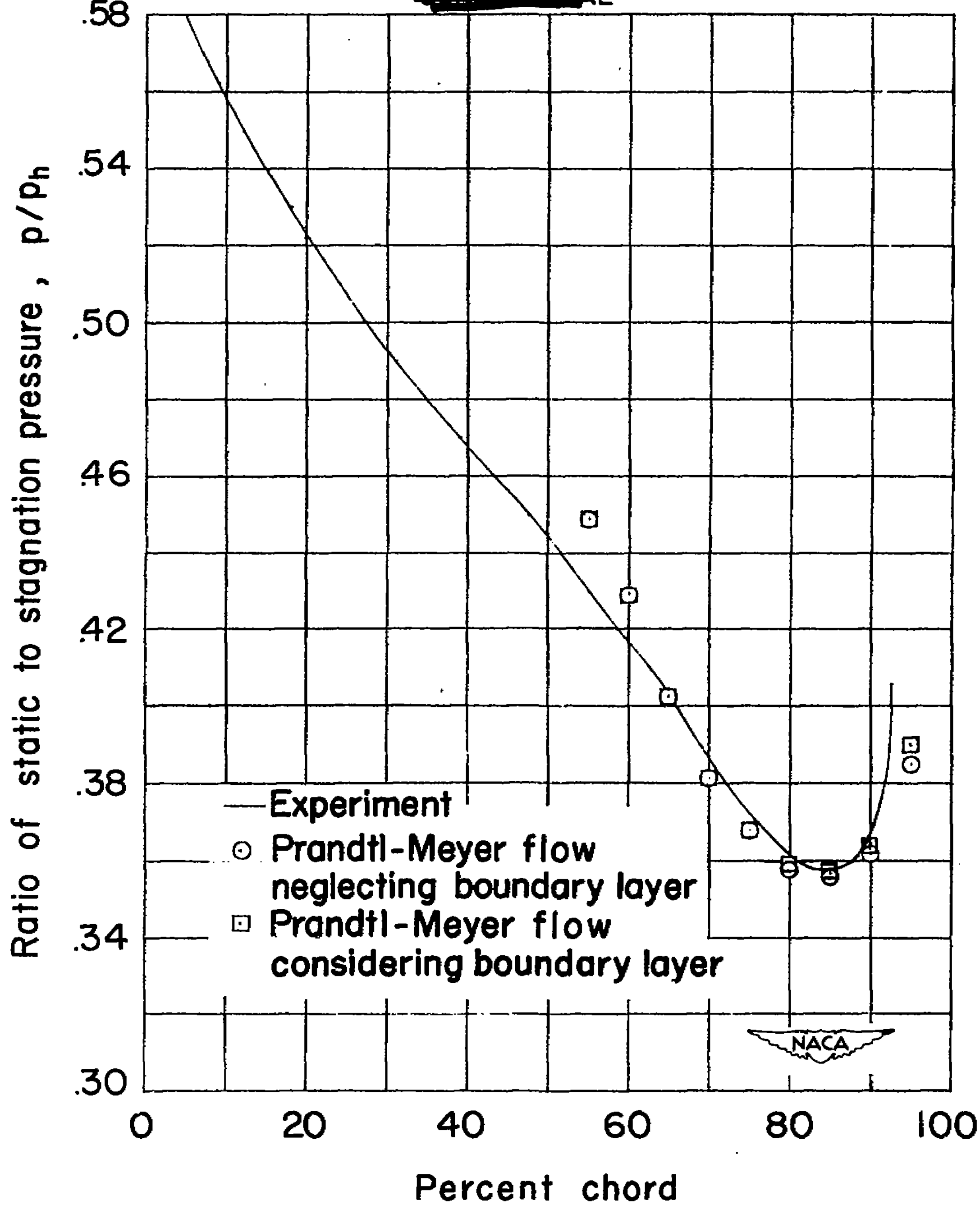

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Figure 7.- Comparison of experimental data with Prandtl-Meyer flow applied from 65-percent-chord station. NACA 66-006 airfoil; $M = 1.00$; $\alpha = 0^{\circ}$.

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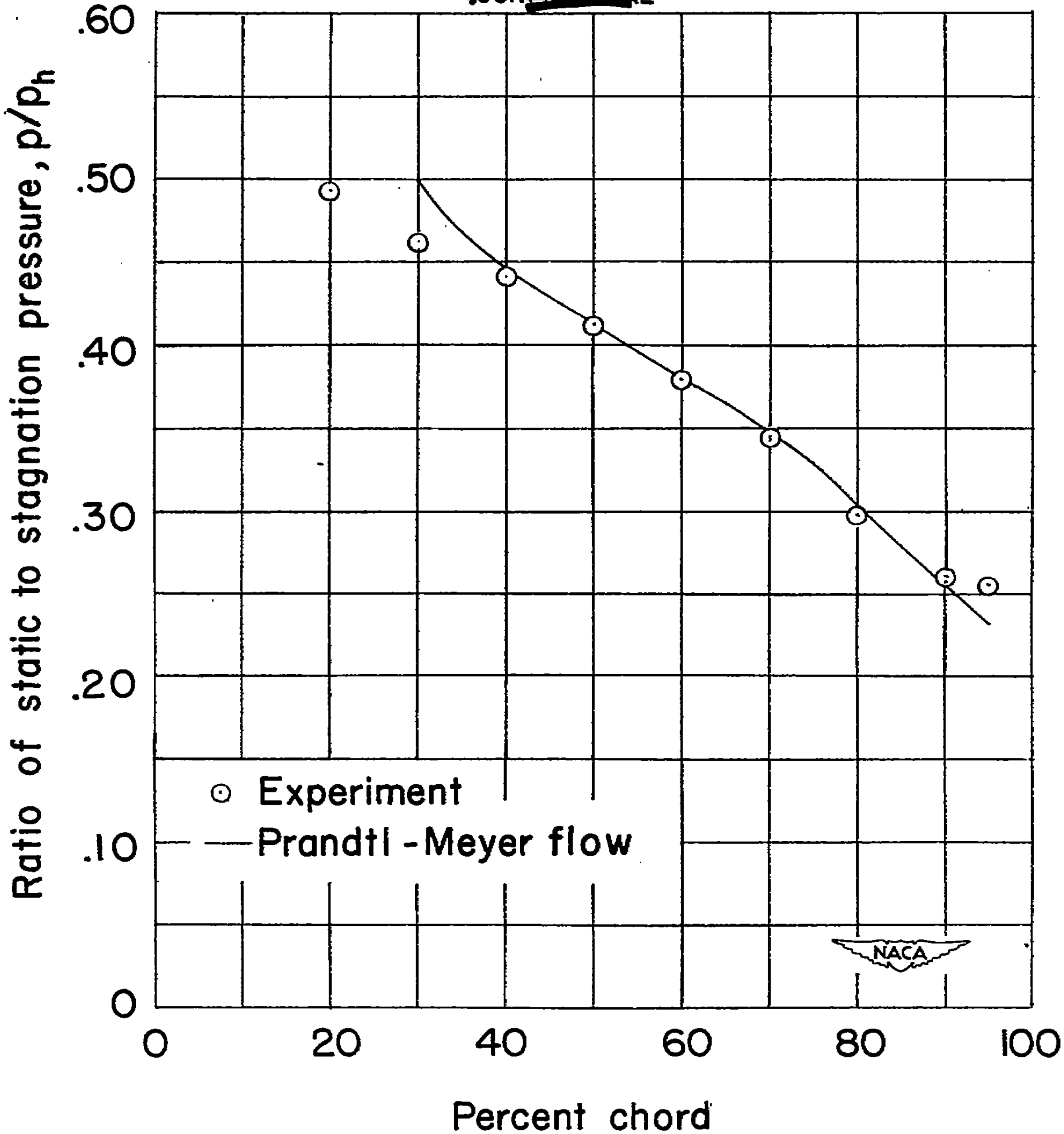

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Figure 8.- Comparison of experimental data with Prandtl-Meyer flow applied from 50-percent-chord station. NACA 16-307 airfoil; $M = 0.984$; $\alpha = -0.04^{\circ}$.

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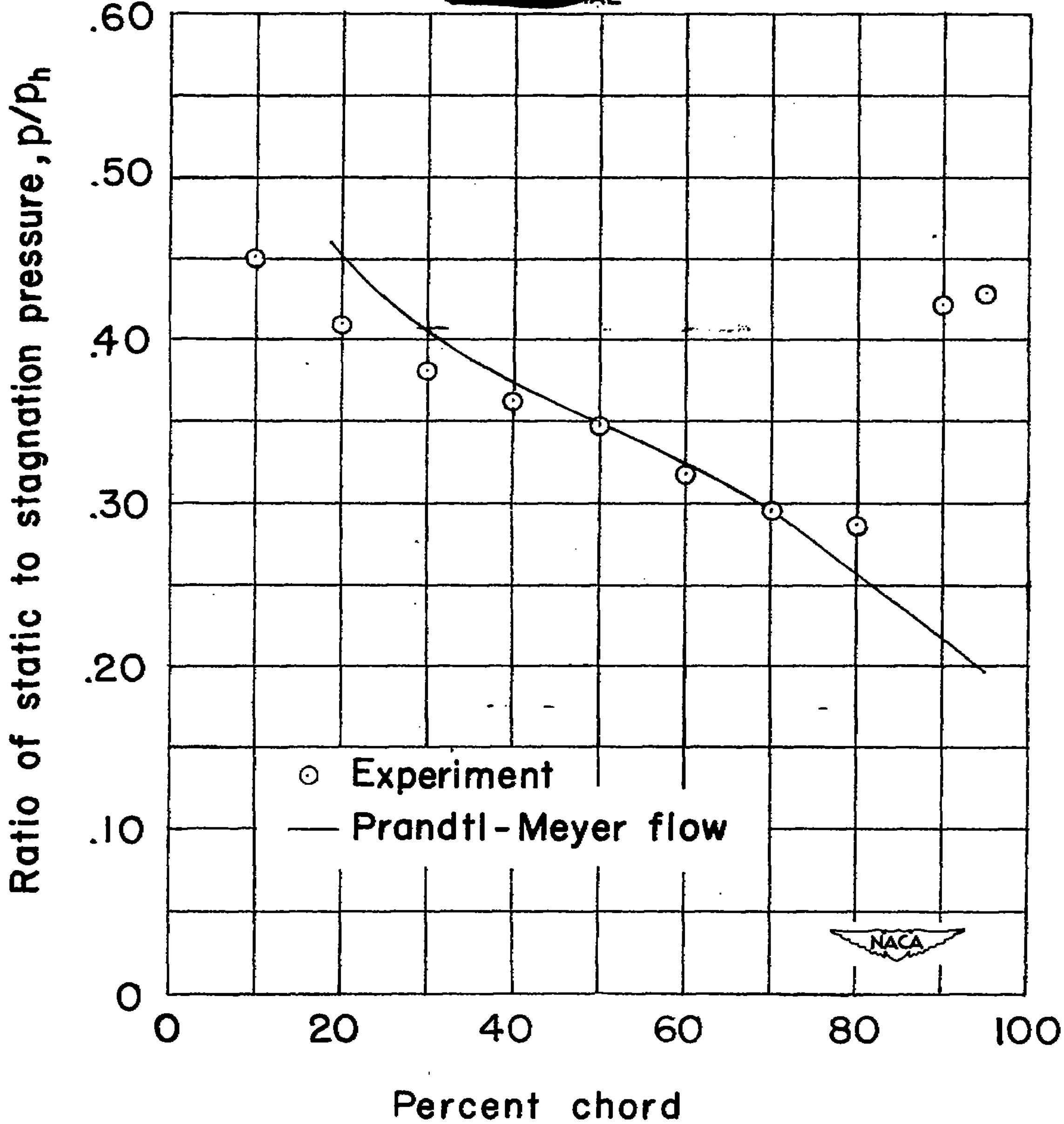

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Figure 9.- Comparison of experimental data with Prandtl-Meyer flow applied from 50-percent-chord station. NACA 16-307 airfoil; $M = 0.955$; $\alpha = 2.71^{\circ}$.

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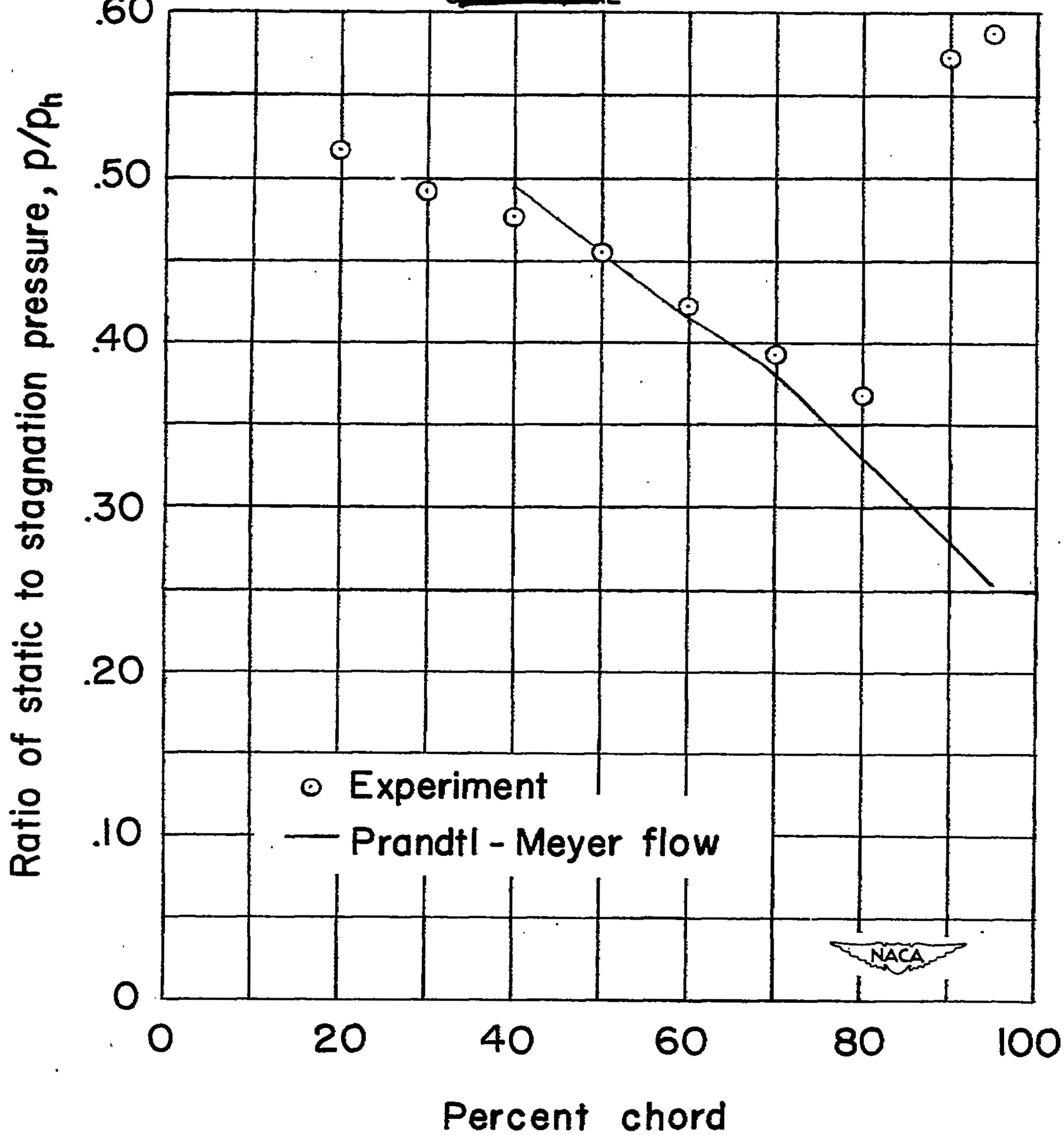

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Figure 10.- Comparison of experimental data with Prandtl-Meyer flow applied from 50-percent-chord station. NACA 16-307 airfoil; $M = 0.882$; $\alpha = 0.54^{\circ}$.

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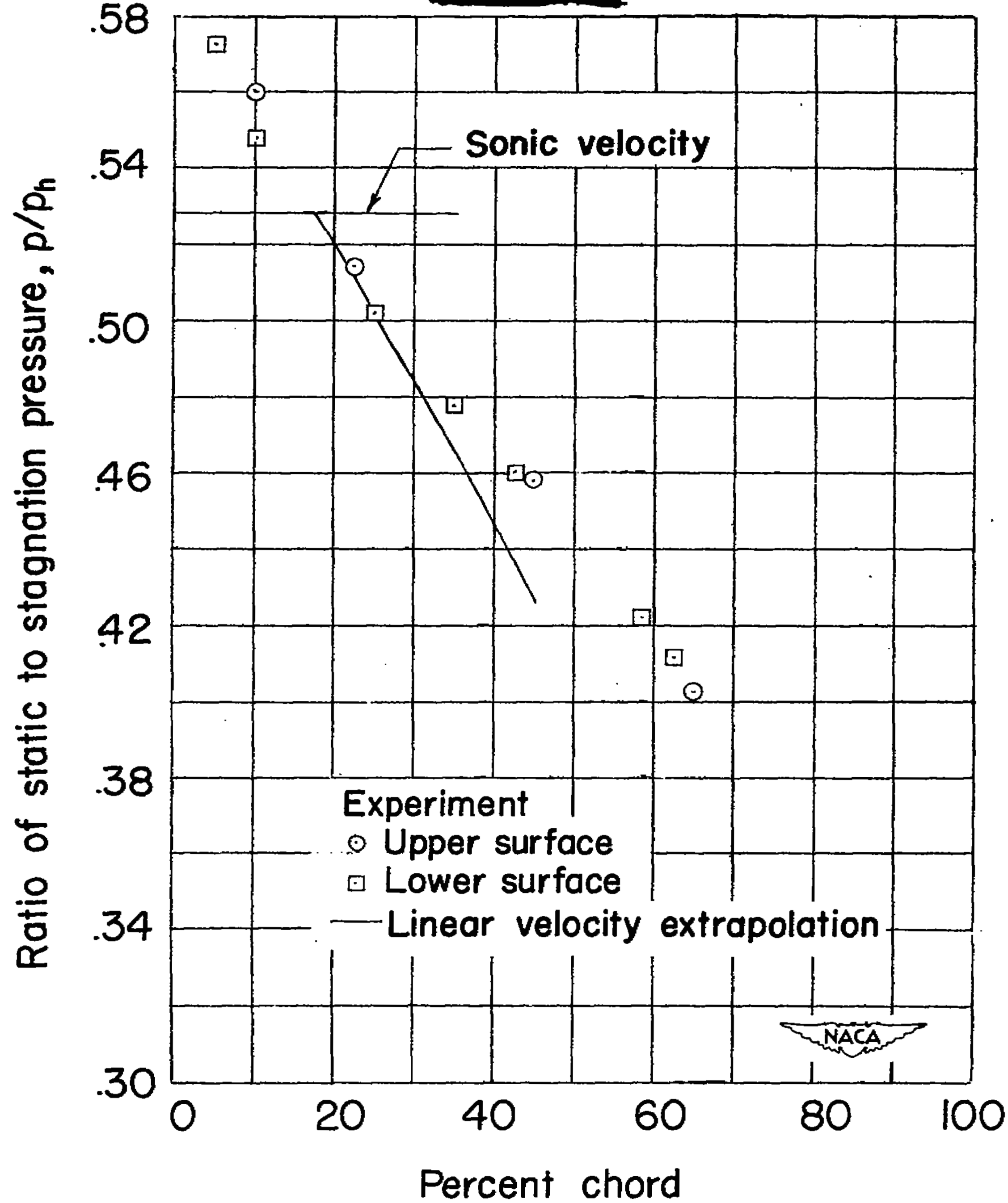

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Figure 11.- Comparison of experimental data with a linear-velocity extrapolation. NACA 66-006 airfoil; $M = 1.00$; $\alpha = 0^{\circ}$.

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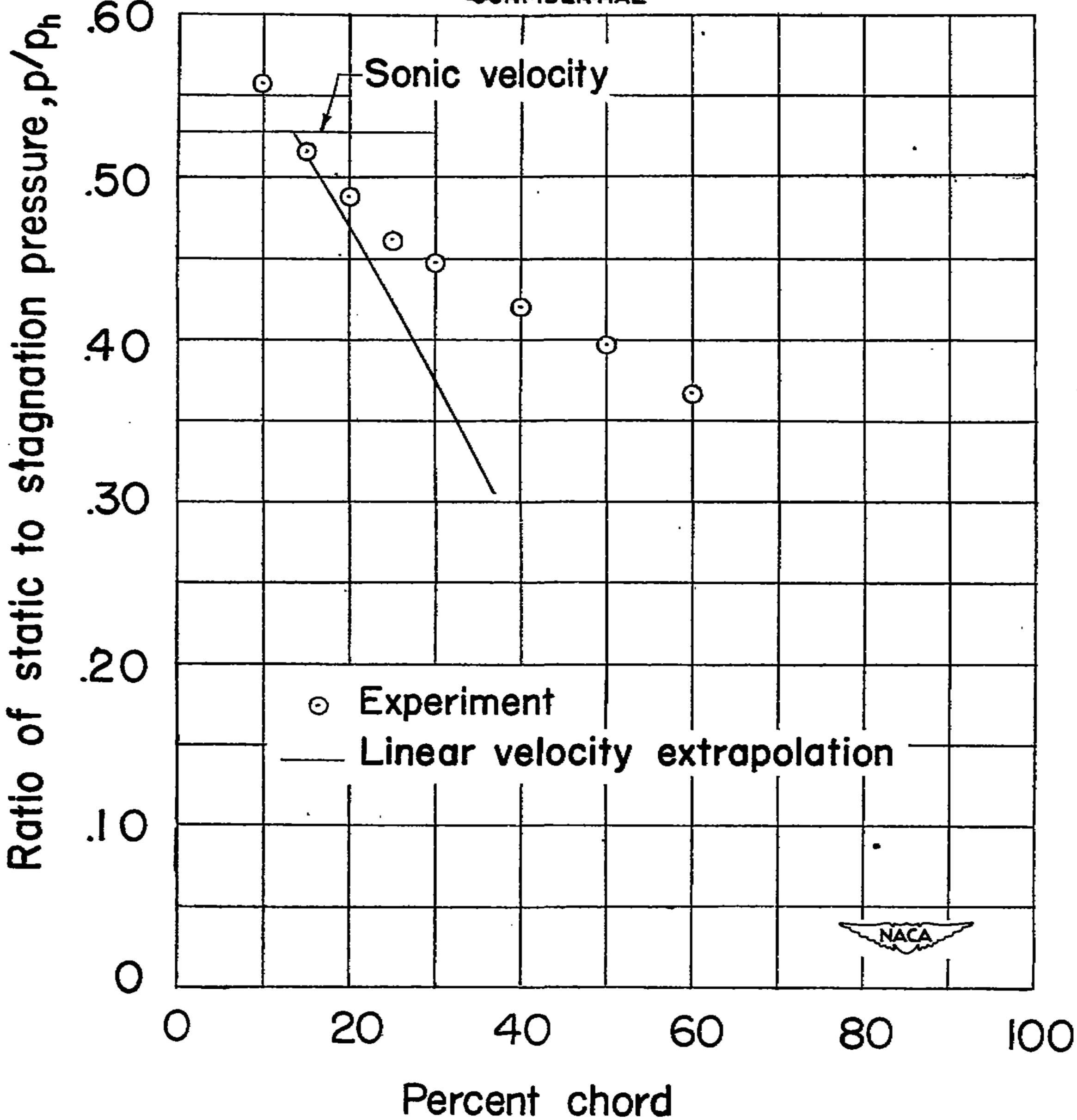

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Figure 12.- Comparison of experimental data with a linear-velocity extrapolation. NACA 16-307 airfoil; M = 1.00; $\alpha = 0.35^{\circ}$.

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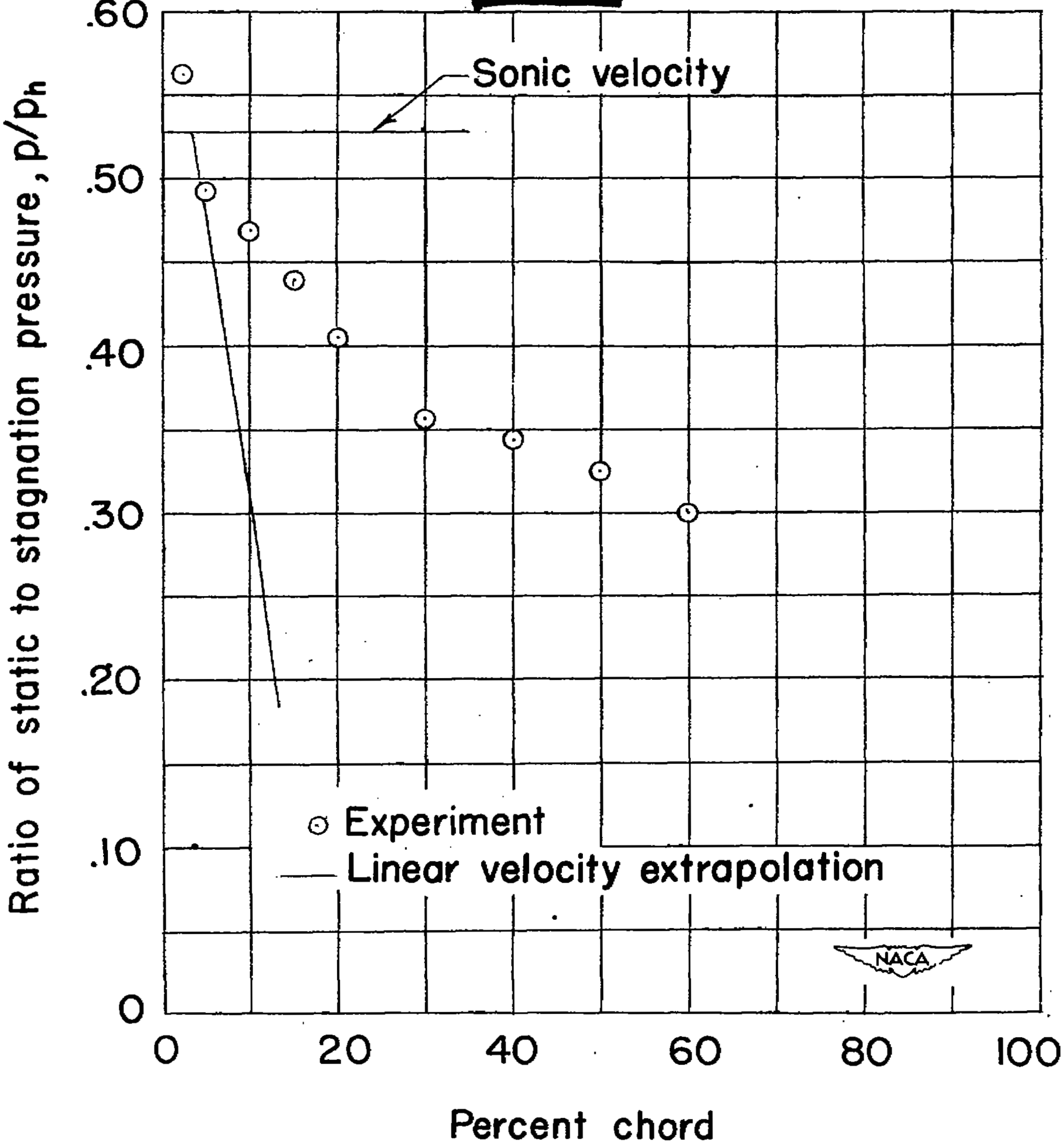

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Figure 13.- Comparison of experimental data with a linear-velocity extrapolation. NACA 16-307 airfoil; $M = 1.09$; $\alpha = 3.0^{\circ}$.

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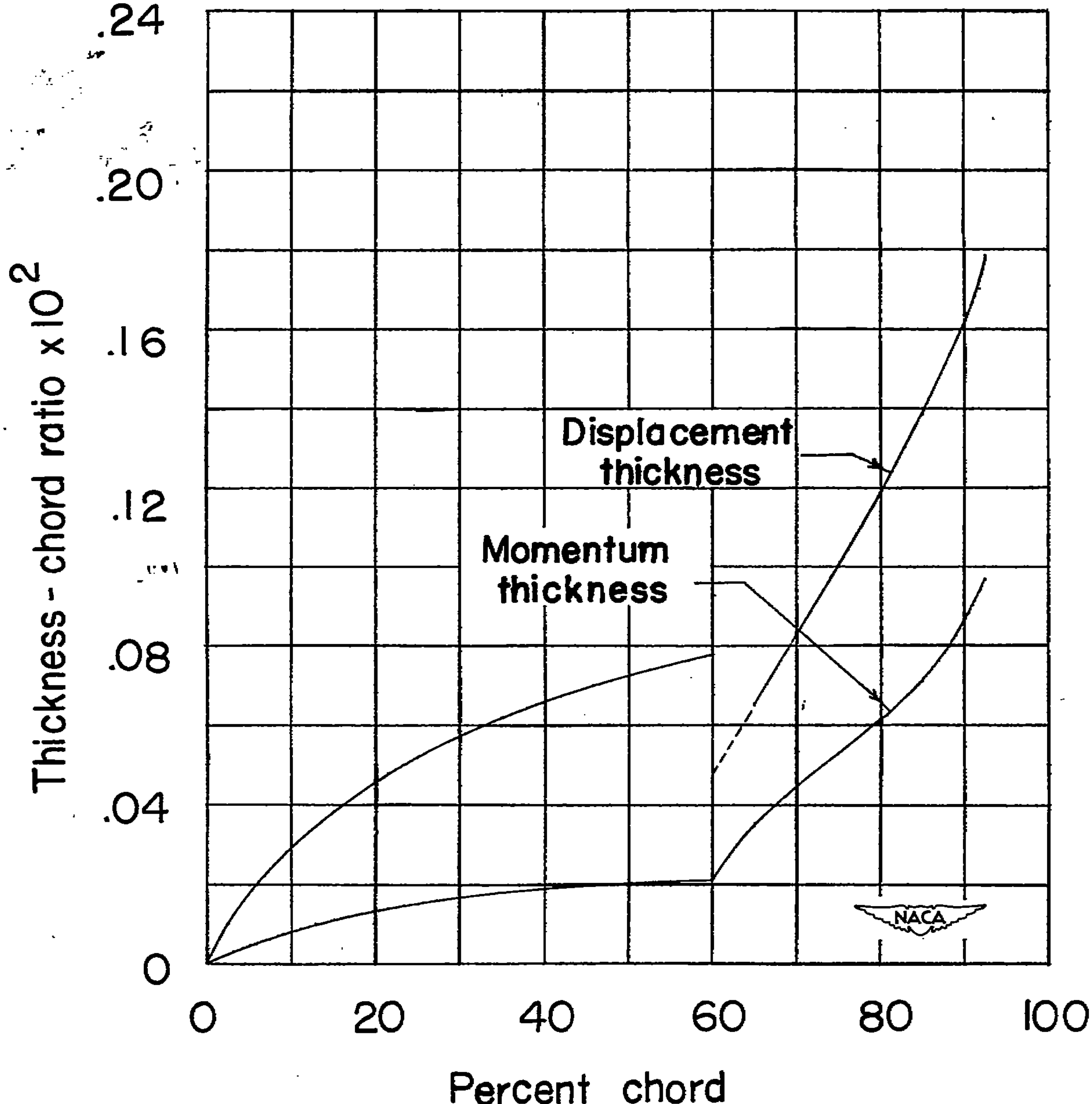

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Figure 14.- Computed boundary-layer growth on an NACA 66-006 airfoil with transition assumed at 60-percent-chord station. M = 1.00; $\alpha = 0^{\circ}$.

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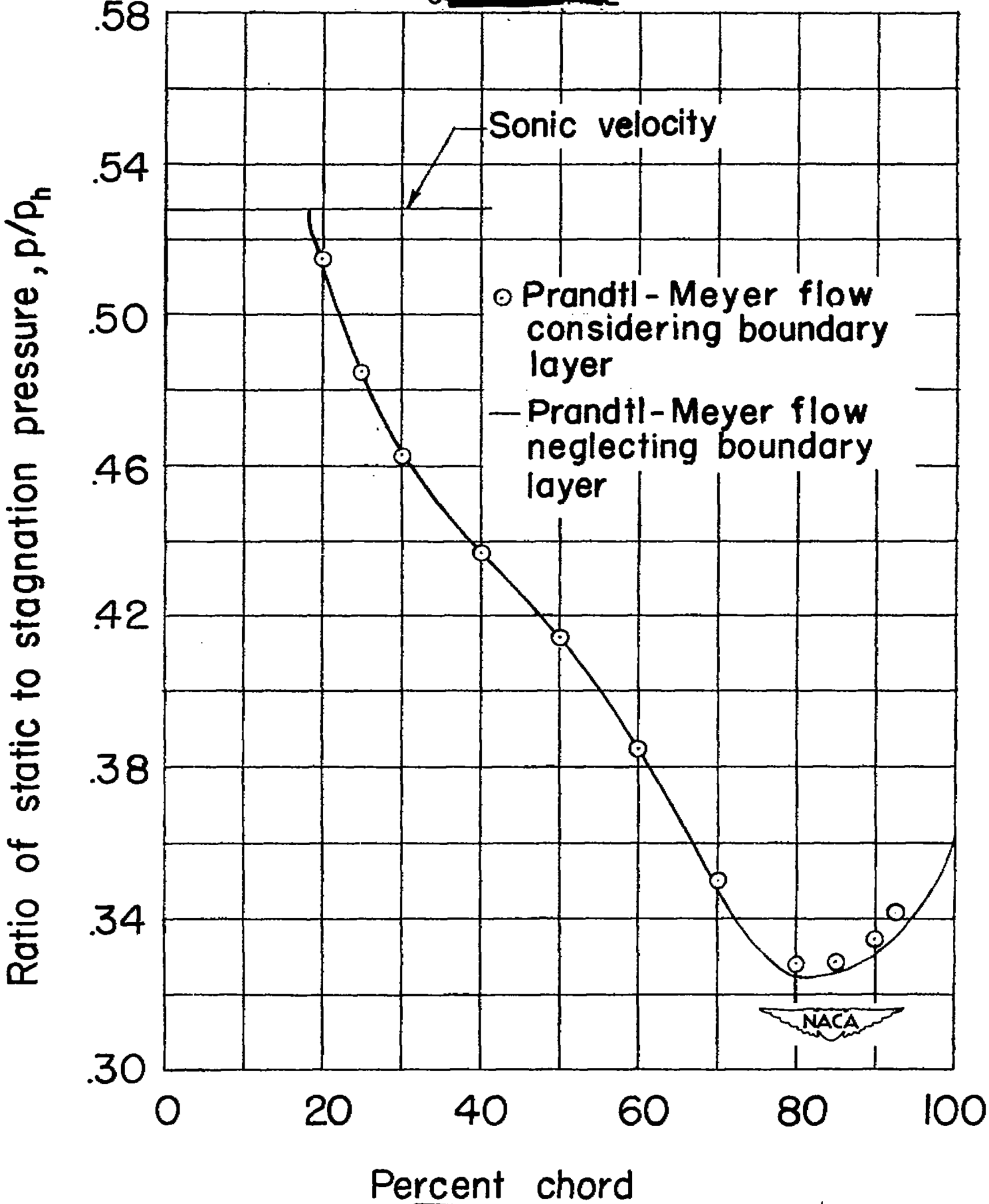

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Figure 15.- Effect of boundary-layer thickness on the pressures predicted by the Prandtl-Meyer flow. NACA 66-006 airfoil; $M = 1.00$; $\alpha = 0^{\circ}$.

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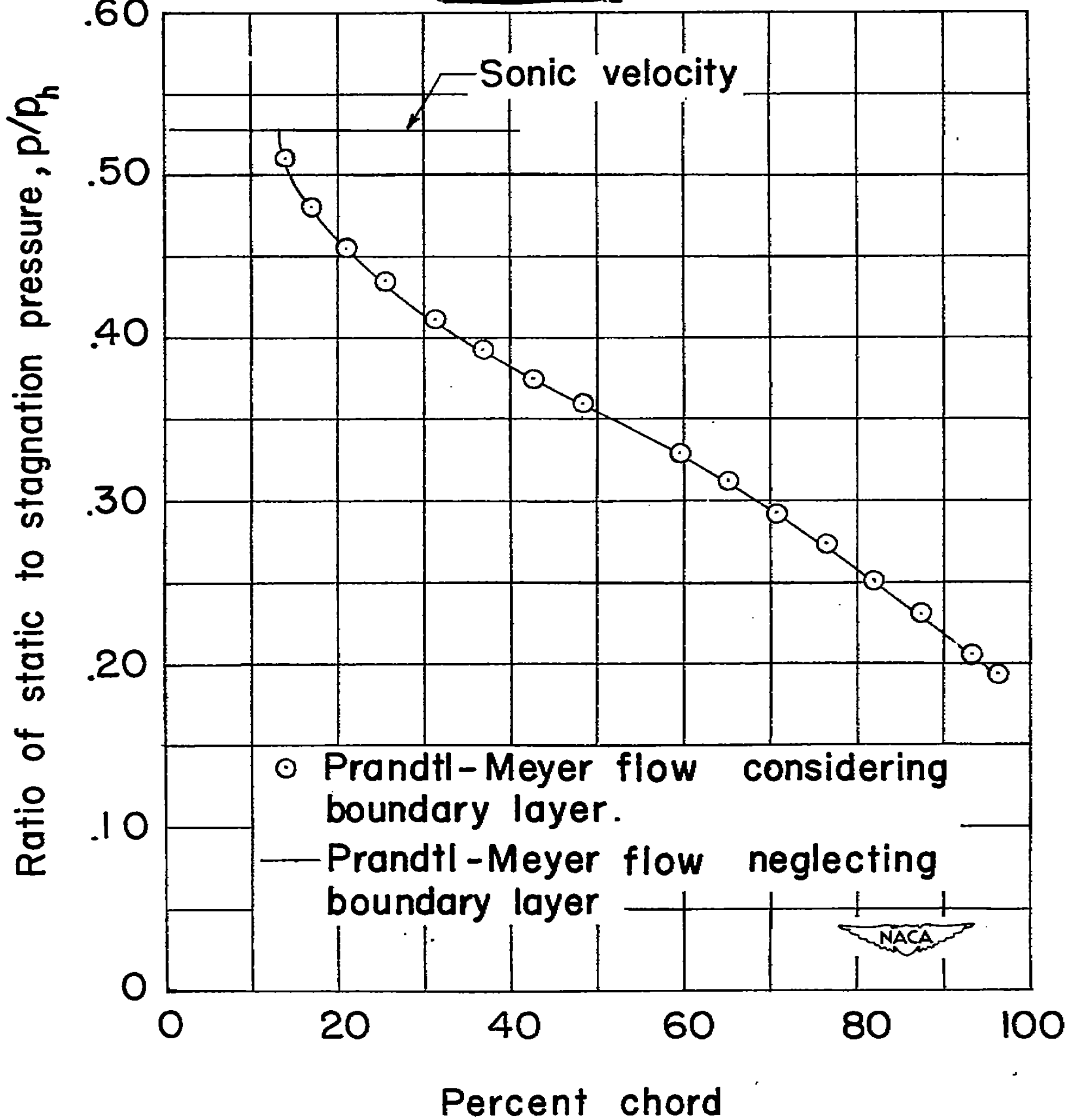

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Figure 16.- Effect of boundary-layer thickness on the pressures predicted by the Prandtl-Meyer flow. NACA 16-307 airfoil; $M = 1.00$; $\alpha = 0.35^{\circ}$.